BELLCOMM, INC.

1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: The Relation of In-flight Loads to the S-II Cryogenic Proof Test
Case 320

DATE: February 9, 1968

FROM: R. E. Hunter

ABSTRACT

A program decision was made recently to cryogenically proof the S-II LH_2 tank to 35.7 psig for all manned flights. This test requires a minimum of new hardware and is actually safer from a strength of material point of view than the present 35 psig room temperature pneumostat test.

This test will proof the tank to all the flight loads except the design wind loads. Recent evaluation of Saturn V design wind loads have shown that significantly lower wind loads are expected without any restrictions on the present 95% launch capability. The cryogenic proof test will envelope flight loads up to 61% of design wind loads.

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MEMORANDUM FOR FILE

I. INTRODUCTION

Recent calculations of Saturn V boost loads have shown that the design wind loads are conservative and that significantly lower in-flight wind loads are expected. The lower values of the expected loads on the S-II LH_2 tank significantly relax the flaw size that must be detected by inspection, in the absence of a cryogenic proof test, to guarantee structural reliability from a fracture mechanics viewpoint. Insufficent experimental data exist to define, without controversy, the maximum allowable flight loads that would allow fracture mechanics to guarantee structural integrity after the present 35 psig room temperature pneumostat test.

A liquid hydrogen proof test of the LH_2 tank for design loads is too difficult to be a practical program alternative, if not actually impossible. The reduced (and more realistic) expected loads make a cryogenic proof test a reasonable and attractive program decision.

A fracture mechanics analysis of the S-II stage LH_2 tank leads to concern about cracks in the weld planes that can survive the room temperature proof test and still cause failure during flight. The significant component of tank stress that can cause relatively small cracks to fail is that component of stress normal to the plane of the crack. A thorough knowledge of the flight loads that produce these stresses is necessary before one can reasonably evaluate the program impact (as well as the structural impact) of this problem. It is convenient to consider loads normal to the longitudinal welds (hoop stress) and loads normal to the circumferential welds (axial stress) separately.

Figure 1 is a sketch of the S-II propellant tanks for reference.

II. LONGITUDINAL WELDS

Figure 2 shows flight stress across the longitudinal welds as equivalent internal tank pressure at station 326 which is near the bottom of the LH_2 tank.

By far the major load contribution comes from the difference between the internal ullage pressure and the ambient pressure. The tank will be pressurized at lift-off to 36 psia maximum (21.3 psig). There will be a 27.5 - 29.5 psig vent valve on S-II-3 (27.0 - 29.0 on S-II-4). Thus when the atmospheric pressure drops to 6.5 psig (about 60 seconds after lift-off) the vent valve will start to vent and maintain a maximum pressure of 29.5 psig to the end of S-IC boost. The hydraulic head of the liquid hydrogen at station 326 under the maximum acceleration of 4.15 g's at S-IC cutoff is about 6.1 psia which increases the internal tank pressure to about 35.6 psi near the bottom of the LH_2 tank. The separate effects of ullage pressure and acceleration head are shown in Figure 2. Note that the present 35 psig room temperature pneumostat test exceeds flight loads for all but a short time near S-IC cutoff.

Longitudinal loads can be reduced by (1) early S-IC cutoff and (2) by a reduction in S-II LH_2 tank ullage pressure. The first possibility can significantly affect payload capability, while S-II LH_2 tank ullage pressure reductions affect the start capability of the J-2 engines for the S-II boost phase of flight. The J-2 engine start capability on the low side of the vent range (27.5 psig for S-II-3) has been verified. Tests are in progress at AEDC to confirm the start capability at 27.0 psig (for S-II-4). It is still not clear if these tests will confirm a lower value.

III. CIRCUMFERENTIAL WELDS

Flight loads on circumferential welds are more complex and less predictable than the loads on the longitudinal welds. This can be seen from figure 3 where circumferential flight loads are shown as equivalent internal tank pressures.

The difference in ullage and ambient pressure again produces a major portion of the circumferential load. The axial compression due to vehicle acceleration, however, reduces circumferential tensile loading. The bending moment at maximum $q\alpha$ produces (an additive) tensile load on one side of the vehicle which is equivalent to an internal tank pressure increase of 21 psig based on design values. From the total load curves and the room temperature proof test, we see that the flight loads can be greater than proof test loads during the maximum $q\alpha$ portion of flight but this is highly dependent on the percentage of design wind loads considered.

The Saturn V design wind is a synthetic profile based on a 95 percentile windiest month omnidirectional wind of 75 m/sec with 99 percentile wind shears and gusts. The design loads were produced by orienting this 75 m/sec wind in the yaw plane to obtain the maximum structural loads. This produced a maximum bending moment on the S-II LH_2 tank circumferential welds of 254×10^6 in lb at station 326.

Recent calculations by The Boeing Company[1,2] in which the synthetic wind profile was not omnidirectional but followed the KSC wind rose have produced wind loads only 72% of design. However, the use of the synthetic profile still overestimates the loads generated by typical wind soundings.[3] Further, the use of annual wind data does not show the full load reduction benefit achieved by the use of a wind biased trajectory.[4] The "expected wind" curve of Figure 3 is an estimate of these conservatisms. If this curve lies below 61% of the design wind curve, the proof test requirements of the circumferential welds are less than those of the longitudinal welds. That is, the circumferential welds can be ignored and the parameters of a cryogenic proof test can be based on longitudinal weld considerations.

IV. CRYOGENIC PROOF TESTING

The comments of the previous sections bear directly on the feasibility of a cryogenic proof test of the S-II LH_2 tank. The design parameters of such a test are discussed in this section.

For a ground test to truly proof a stage for flight loads it must not only apply the flight stress levels but even exceed them by a sufficient amount to allow for flaw growth from cyclic and/or sustained stress from the time of the ground "proof" test up to and including flight.

The proof test criteria used for the Saturn V launch vehicle propellant tanks has been[5]:

Parent Metal:

proof stress = 1.05 x flight stress but not to exceed 0.95 x yield stress

Weld Metal:

ultimate factor of safety during proof $\geq$ 1.20

IT SHOULD BE EMPHASIZED THAT NO DATA EXIST ON S-II WELDMENT SUSTAINED-STRESS FLAW GROWTH AND VERY LITTLE DATA EXIST ON CYCLIC FLAW GROWTH. A 1.05 PROOF FACTOR APPEARS ADEQUATE BUT FURTHER DATA ARE NEEDED TO FIRM UP THIS NUMBER.

A. Longitudinal Welds

To satisfy the 1.05 x maximum flight stress on the longitudinal welds would require a proof pressure of 35.8 psig derived as follows:

Max. flight load at station 326 (equivalent tank pressure)	35.6
Proof factor	x 1.05
Proof pressure at station 326	37.30
Static head of LH_2 under 1g (902-326) x (.00254) x (1.g)	- 1.50
Required ullage pressure	35.80 psig.

B. Circumferential Welds

Using the Boeing predicted[1] circumferential loads, a proof pressure of 38.2 psig is required as shown below:

Max. flight load at station 326 (equivalent tank pressure)	36.4
Proof factor	x 1.05
Required ullage pressure	38.20 psig

Note that a significantly higher pressure is required to proof the circumferential welds (38.2 psig) than the longitudinal welds (35.8 psig) if the Boeing wind load predictions are used.

C. Stage Capability

Since the S-II stage design is based on cryogenic strengthening, it is pertinent to first look at what pressures are attainable in the S-II LH_2 tank without violating the previously stated proof test criteria.

Aluminum alloy 2014-T6 is about 35% stronger at -423°F than at room temperature; the weldments are about 27% stronger. If a cryogenic proof test is performed with the LH_2 tank filled with liquid hydrogen to station 902 (see Figure 1) the entire LH_2 tank below station 902 will be wetted with LH_2 and therefore at -423°F. Only the forward dome of the LH_2 tank will be significantly warmer than -423°F. Thus the temperature that can be maintained in the forward dome will govern the allowable pressure in the LH_2 tank during such a test.

Figure 4 shows the various proof test design parameters. The two solid curves represent the two previously stated criteria for welds and parent metal. Note that for all temperatures the parent metal yield criterion is more severe than the weld ultimate factor of safety criterion. That is, for a given temperature, the yield criterion demands a lower pressure than the weld ultimate factor of safety criterion.

D. Proof Test Parameters

Due to higher risks associated with a cryogenic proof test from a procedural point of view, a new criterion has been considered to maintain a 1.2 ultimate factor of safety in the parent metal. Applying this criterion to the parent metal in the forward dome of the LH_2 tank produces the dashed curve of figure 4. Above -360°F this is the most severe criterion. Below -360°F the parent metal yield criterion is more severe.

Temperatures in the forward dome may be as high as -210°F using present equipment at MTF. Recent tests on S-II-4 at MTF measured -240°F in the forward dome of the S-II LH_2 tank when filled to station 902 with liquid hydrogen. To guarantee temperature lower than this range would require additional equipment (heat exchangers) at MTF.

Ruling out temperatures below the -210°F to -240°F range, we see from figure 4 that if the parent metal ultimate factor of safety is to be maintained at or above 1.2, the maximum allowable pressure would be 37.2 psig. Allowing 1.5 psig for uncertainty in vent valve cracking pressures and instrumentation tolerances, a cryogenic proof test can be run that guarantees a pressure of 35.7 psig. This is only .1 psig less than the calculated value necessary to proof the longitudinal welds. If the ullage pressure at the end of S-IC boost is near the reseat value of the vent valve (rather than at the maximum value used to calculate a cryogenic proof pressure of 35.8 psig) the 35.7 psig cryogenic proof test should be adequate to proof the longitudinal welds to a minimum of 1.05 x (max. flight loads).

Note that the current 35 psig room temperature proof test produces 95% of yield and a 1.19 parent metal ultimate factor of safety. From a conventional strength of materials point of view, the room temperature test is more hazardous than the proposed cryogenic test!

A proof test to 35.7 psig will only proof the circumferential welds to 61% of design loads or to 85% of the Boeing-predicted loads. Additional ullage pressure reductions of 2.4 psig would be necessary to include the Boeing predicted wind loads within the 35.7 psig cryogenic proof test. This would mean that the J-2 engine must start at a pressure as low as 25.1 psia (27.5 - 2.4 = 25.1 psia). It is more reasonable to expect less than 61% of design loads due to the conservatism still remaining in the Boeing predicted maximum loads.[5] This does not, however, deny the fact that additional ullage pressure reductions increase the structural margin of safety of the S-II LH_2 tank.

V. SUMMARY

An LH_2 proof test to 35.7 psig appears reasonable from the test parameters. The adequacy (or pedigree) of such a test depend on additional data which include:

1. Sustained stress and cyclic flaw growth data of 2014-T6 aluminum weldments,
2. Realistic prediction of upper bound wind loads,
3. Minimum NPSH requirements for J-2 engine start.

The status of item 1 is unknown. Item 2 is currently under intensive investigation.[5] Tests at AEDC to provide data necessary for item 3 have been requested[6,7] but the actual planning is not known.

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2031-REH-mch

R. E. Hunter

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7. NASA Headquarters TWX MAT-4, Subject: J-2 Engine Testing at AEDC, January 24, 1968.

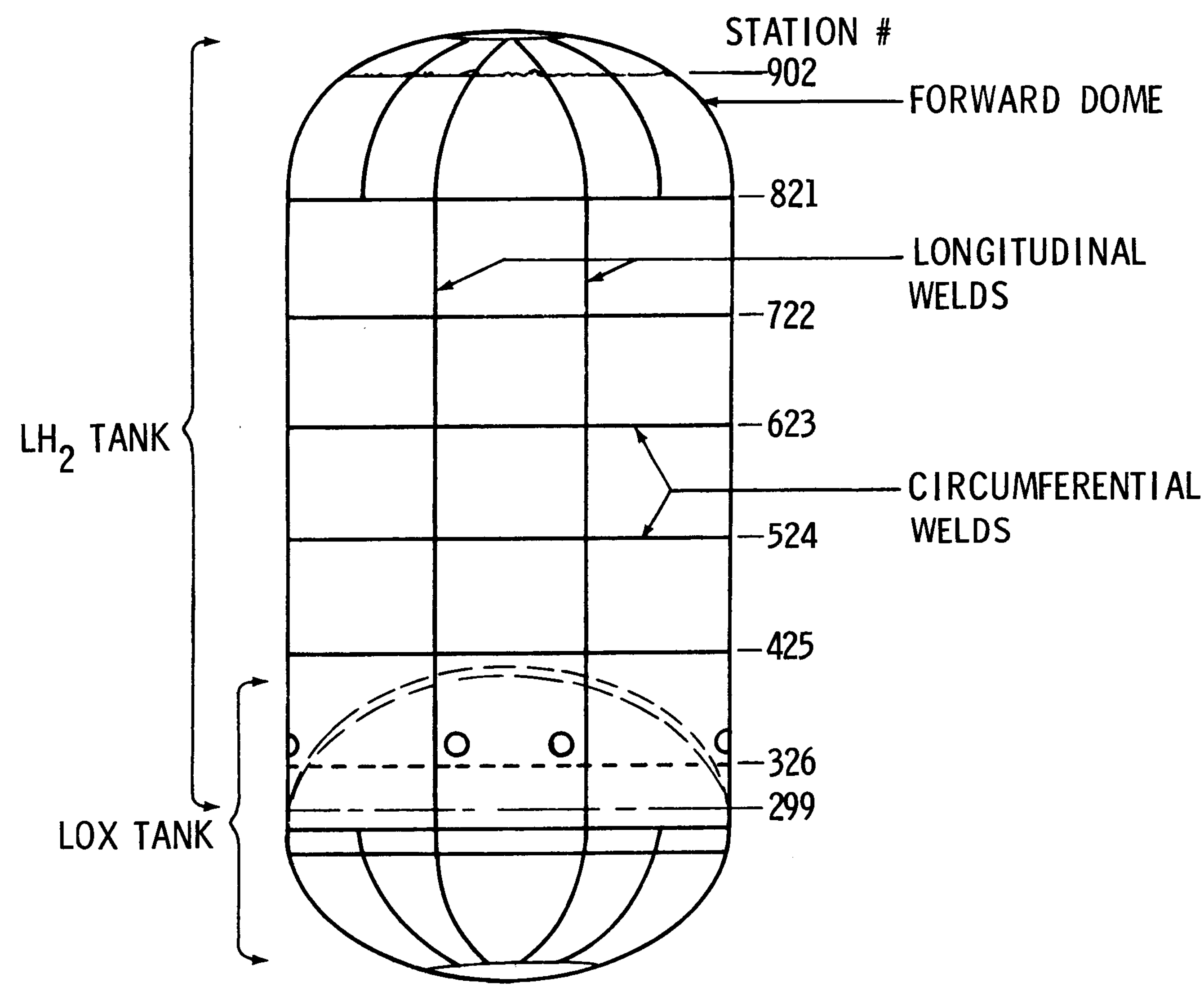

FIGURE 1 - S-II PROPELLANT TANKS

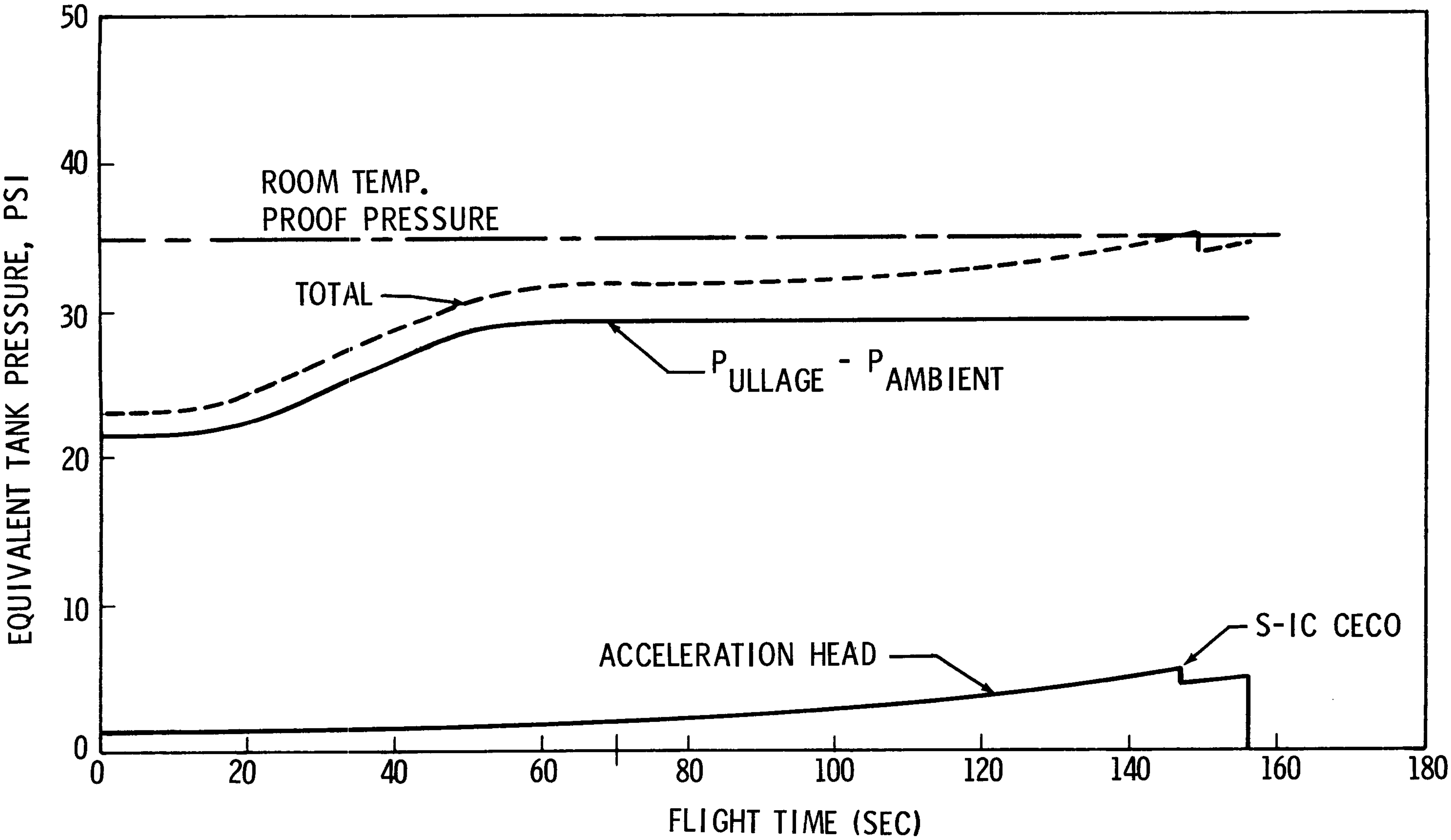

FIGURE 2

S-II LH_2 TANK LONGITUDINAL LOAD AT STA 326
(IN TANK PRESSURE EQUIVALENT)

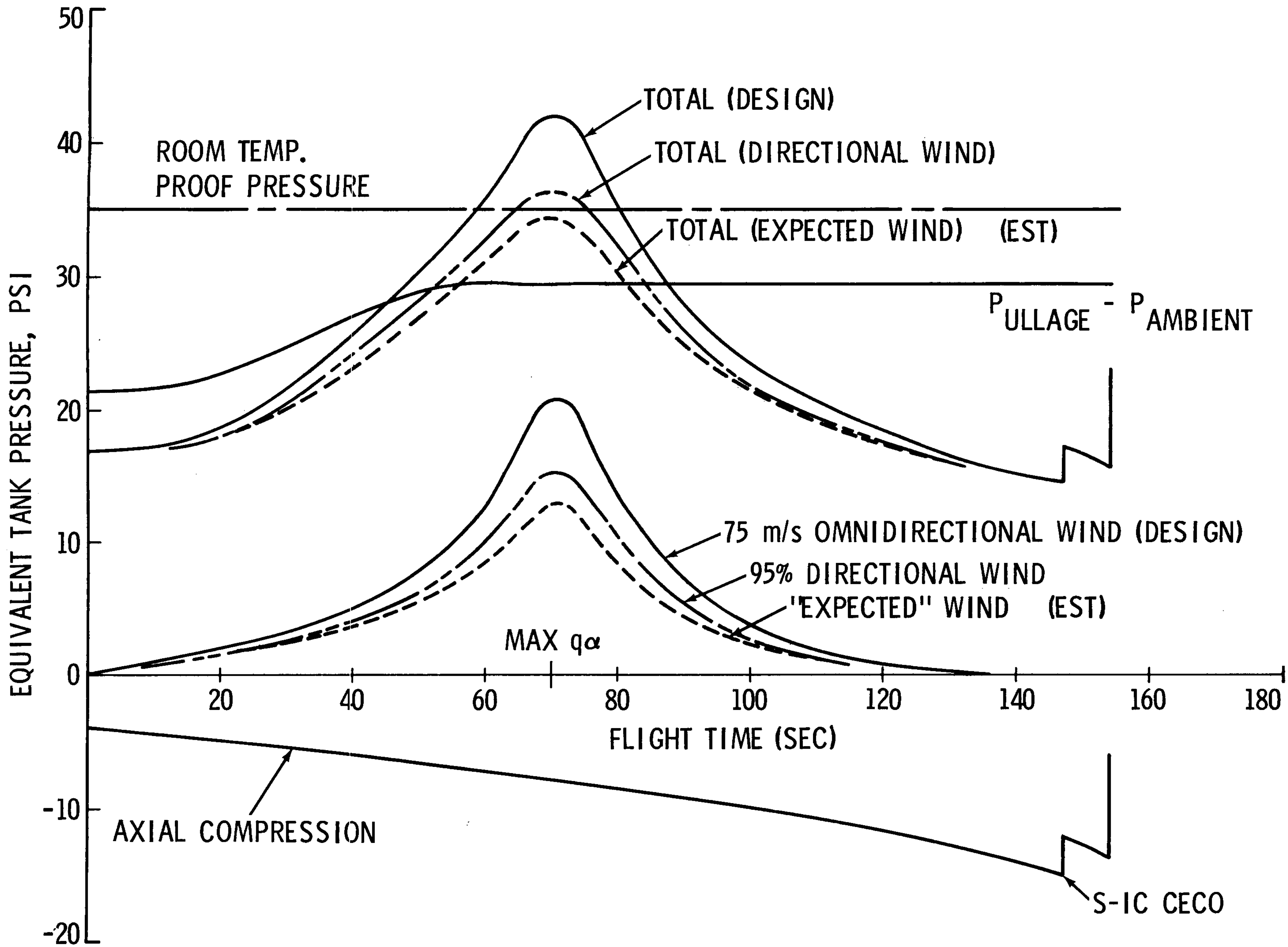

FIGURE 3

S-II LH_2 TANK CIRCUMFERENTIAL LOAD AT STA 326
(IN TANK PRESSURE EQUIVALENT)

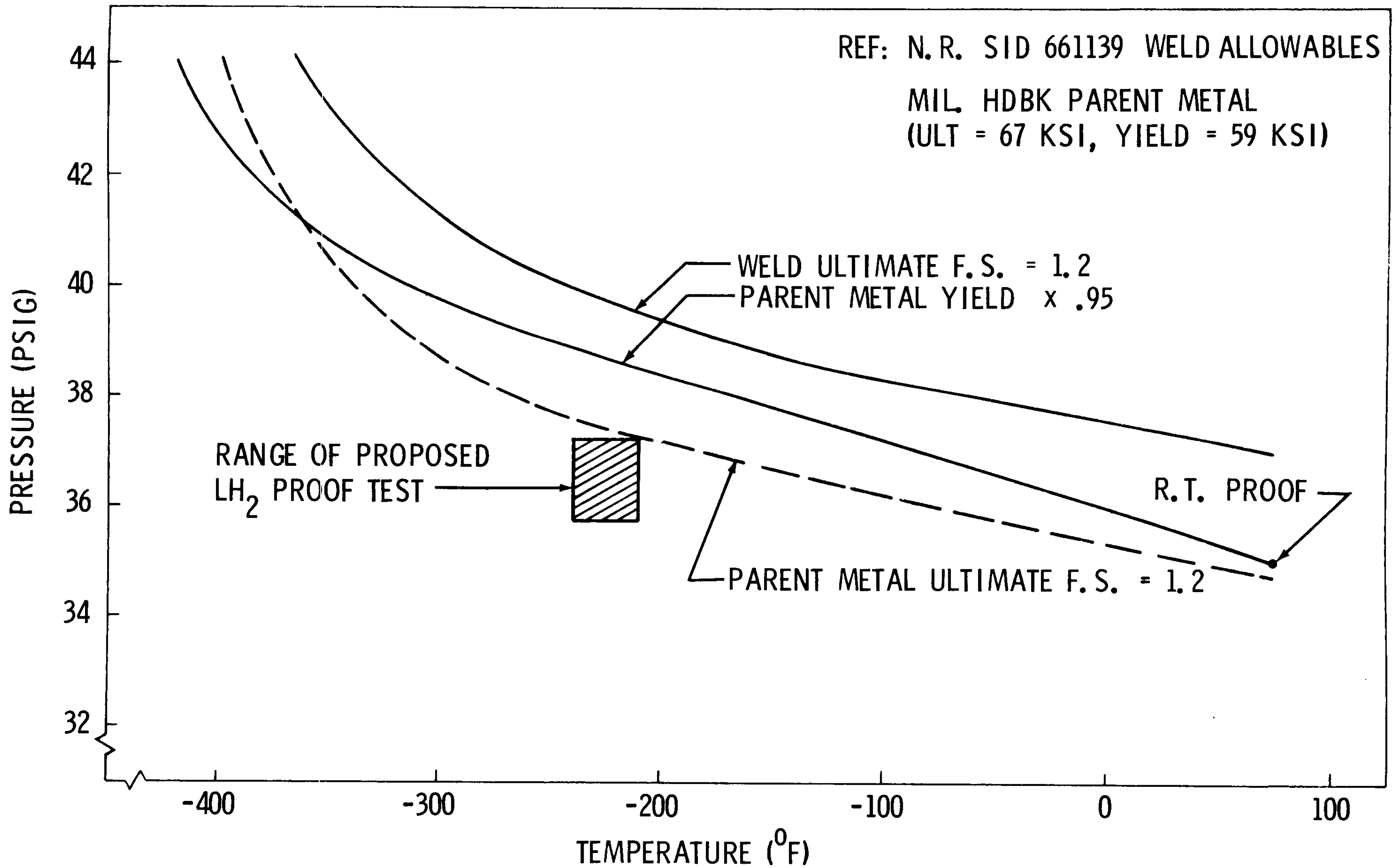

FIGURE 4 - PROOF TEST DESIGN PARAMETERS

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